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# Low-Pressure Performance of Annular, High-Pressure (40-atm), High-Temperature (2480 K) Combustion System

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# Low-Pressure Performance of Annular, High-Pressure (40-atm), High-Temperature (2480 K) Combustion System

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## Summary

Experimental investigations have been conducted to develop a combustion system that would furnish combustion gases for a high-pressure, high-temperature turbine-cooling research facility. The maximum turbine conditions desired were about 40 atmospheres gas pressure and 2480 K gas temperature.

The present investigation was conducted at pressures to 7 atmospheres, 589 and 894 K inlet-air temperatures, and 2365 K exhaust-gas temperature. At a 589 K inlet-air temperature a maximum combustion efficiency of about 99 percent was attained at a 0.036 fuel-air ratio; the efficiency decreased to 93 percent as the fuel-air ratio was increased to 0.056. At a fuel-air ratio of 0.02 the efficiency was about 96 percent. At 894 K inlet-air temperature combustion efficiencies were virtually 100 percent to a fuel-air ratio of 0.045 and then decreased as the fuel-air ratio was further increased to 0.053.

Tests to determine the isothermal pressure loss of the combustion system show a liner loss of 1 percent with a 6 percent system loss.

Unburned hydrocarbon index values were below 1.0 over the fuel-air ratio range with 894 K inlet-air temperature. A minimum carbon monoxide index value of about 3 was attained at a fuel-air ratio of 0.034.

An average liner metal temperature of 1204 K, 310 kelvins greater than the inlet-air temperature, was reached with an 894 K inlet-air and a 2365 K average exhaust-gas temperature. The maximum local metal temperature at this condition was 515 kelvins above inlet-air temperature.

## Introduction

The performance characteristics of an experimental combustion system designed for high-temperature operation at 40 atmospheres pressure are documented at pressures of about 7 atmospheres. The combustor will be used as the hot-gas source for a high-pressure turbine-cooling research facility at the Lewis Research Center. This report covers the performance of this combustor at exhaust-gas temperatures to 2365 K, with a nominal inlet-air temperature of 894 K.

The compressor pressure ratios and turbine-inlet temperatures of aircraft gas turbine engines have been increasing steadily since the introduction of these engines during World War II. In both commercial and military applications the trend is toward turbofan engines with compact, high-pressure, high-temperature gas generators. Cycle and mission analyses, such as reported in reference 1, have indicated potential compressor pressure ratios as high as 40 for future engines with turbine-inlet temperatures to 2480 K. In these analyses it was assumed that combustors and turbines can be developed to operate satisfactorily at these high pressure and temperature levels although design criteria are nonexistent.

The higher compressor pressure ratios will be accompanied by higher compressor discharge temperatures. As a consequence, the compressor discharge air, which is the coolant for the combustor and the turbine, will have lower heat-sink capacity. At the same time, the cooling requirements will increase because the combination of high pressure and high temperature will result in higher heat fluxes to engine parts exposed to the gas-path environment. These high heat fluxes will impose such problems as high temperature gradients through the turbine vane walls, as discussed in reference 2. These temperature-gradient, heat-transfer, and combustion problems generated in the high-energy gas paths of advanced engines cannot be properly evaluated by extrapolating data obtained in the lower-energy gas paths of current engines or test rigs. An engine or a test rig designed for high-pressure, high-temperature operation is required for such an evaluation. Simulating such an environment in a specially designed test rig is the most practical way to provide for this evaluation.

To achieve this simulation, the NASA Lewis Research Center has built a new facility consisting of independent and parallel combustor and turbine-cooling test sites with supporting service systems, as described in reference 3. An integrated system of minicomputers and programmable controllers will provide automated control, safety monitoring, and data acquisition for the entire facility. The turbine-cooling research facility requires a combustor mounted upstream of the turbine to provide exhaust gases to a temperature of 2480 K.

Combustor development and performance tests were conducted at low pressures (5 to 7 atm) and at inlet-air temperatures of 589 and 894 K. Exhaust-gas temperatures were steadily increased, with considerable testing done at exit temperatures in excess of 2150 K. Combustor performance and

emissions results are presented in this report for the final version of the combustor.

## Apparatus and Procedure

### Test Facility

The full-annular combustor was tested in a closed-duct test facility connected to a laboratory air supply and exhaust system. Airflow rates and combustor pressures were regulated by remotely controlled valves upstream and downstream of the test section. Flow straighteners in the ducting evenly distributed the air entering the combustor. For these tests the combustor-inlet air was heated to nominal values of 589 and 894 K without vitiation. Maximum air pressure available was about 10 atmospheres. Additional facility capabilities are described in detail in reference 4.

### Research Hardware

*Test rig.*—Reference 5 presents data from tests with a particular type of combustion system that uses single-plane fuel injection with modified air-blast fuel modules. The results showed good efficiency at exhaust-gas temperatures as high as 2400 K. This type of hardware was used as a basis for preliminary design of the hardware for the investigations reported herein. Figure 1, a cross-sectional view of the combustion test rig, shows instrumentation planes and various dimensions. The inlet centerbody contour increases the inlet airflow velocity to simulate the air discharge from an axial-flow compressor. As shown in figure 2 the air enters a dump diffuser, from which about 80 to 85 percent of the air enters the air-blast fuel modules and the remainder flows along the inner and outer combustor annuli and through the film-cooling holes in the combustor liner. There is no addition of dilution air to the combustion chamber. The combustion gases pass through an annulus area that represents the turbine-inlet area and are then cooled by water sprays before they enter a facility exhaust system.

*Fuel modules.*—Fuel is introduced into the combustion system by two rows of air-blast fuel modules. Figure 3(a) shows the air and fuel flow through one of these modules. Fuel for each module is supplied by a fuel tube with a 0.098-centimeter-diameter discharge orifice. The fuel impinges on a splash plate that injects the fuel into the swirling air. Since the fuel impinges on the splash plate downstream of the center or inner air swirler, there is no fuel-air mixture in the module body to ignite by flashback or autoignition. See figure 3(a).

The air swirlers of each fuel module consist of two annular swirlers with vanes positioned to give counterswirl. All vanes are at an angle of 45° to the axial direction. Figure 3(b), a view looking upstream, shows swirl directions and some dimensions. Note that the outer swirler of both the outer row and inner row of fuel modules tends to corotate the air at the interface between the two rows. Figure 4 shows the two rows of fuel modules with the inner combustor liner installed. Also shown are the outer-row module supports and the blocking plate between modules.

*Combustor liner.*—The separate inner and outer liners are shown in figure 5. The two liners, when fitted together, form an annular combustion zone, as shown in figure 2. Each liner is constructed of seven circumferential overlapping panels. Cooling air is directed onto the back side of the trailing edge of a panel by a row of film-cooling-air holes and is then discharged along the hot-gas side of the next downstream panel in the form of a cooling film. The combustor liner overall length is 23.9 centimeters, and the length from the fuel-module discharge plane to the exhaust instrumentation plane is 22.0 centimeters. The maximum cross-sectional area (reference area) of the combustor annulus is 0.2474 square meter.

Detailed thermal, stress, and film-cooling analyses were made with 1260 K as the maximum allowable liner metal temperature. Results indicate that Hastelloy X with a thickness of about 0.20 centimeter is a suitable liner material. The analyses were made for uncoated metal surfaces. Before actual tests the liners were coated with a thermal-barrier coating. Reference 6 gives details of the coating and the method of application.

The liners were film-airflow calibrated at ambient pressure and temperature to determine the validity of assumptions made for calculating the film-cooling-air flow rate. A sketch of the calibrating stand is shown in figure 6. Each row of holes could be calibrated separately, or the entire liner could be calibrated with all the rows unblocked. Calibration results are listed in table I.

*Instrumentation.*—Combustor inlet-air instrumentation was mounted at planes 2 and 3, as shown in figure 1. Figure 7(a) shows position details of the inlet-air Chromel-Alumel thermocouples. Figure 7(b) shows the dimension details of the eight total-pressure rakes of four probes each, installed at centers of equal areas, and of the 16 wall static-pressure taps, eight equally spaced around both the inner and outer walls. The indicated readings of the inlet-air thermocouples were taken as true values of the total temperature.

Exhaust-gas instrumentation was mounted at plane 4, as shown in figure 1. Dimension details are given in figure 7(c). There were eight fixed rakes that

each contained five probes for total pressure and five Pt/Pt-13% Rh aspirated thermocouples, the readings of which were taken as true values of total temperature. All were mounted at centers of equal areas. Static pressure was measured by four wedge static-pressure probes equally spaced around the annulus at area centers. Four gas sample rakes, each with three area-centered probes, were equally spaced around the circumference. The rakes were plumbed so that gas samples could be obtained from any individual rake or from combinations of two, three, or four rakes. The three probes of each rake were all tubed to a common manifold.

Airflow rates were measured by a square-edged orifice installed according to ASME specifications. Fuel flow rates were measured by turbine flowmeters with frequency-to-voltage converters for readout and recording. The fuel was ASTM Jet-A.

Chromel-Alumel thermocouples were installed on both combustor liners to obtain a measure of liner average metal temperature. The thermocouples were installed so as to indicate the hot-gas-side surface temperature of the metal, the side on which the thermal-barrier coating had been applied. There were 13 thermocouples on the outer liner and 12 on the inner liner. Eight were equally spaced around liner panel 4, straddling centerlines (fig. 5); and one each was placed at 0° and 180° on panels 1 and 7. The outer liner had one additional couple at 180° on panel 3. Panel 4, on which the eight circumferential thermocouples were installed, is near the axial plane of the maximum flow area, or reference area.

### Test Conditions and Procedure

The facility used for these tests had a maximum air pressure capability of about 10 atmospheres. Since the combustion system is to be operated at pressures as high as 40 atmospheres, the inlet conditions were adjusted to give diffuser-inlet Mach numbers similar to those that would be obtained in the 40-atmosphere facility. The various test conditions are listed in table II. Also listed are film-cooling-air flow rates.

Combustor performance data were obtained by setting the controls for a particular inlet condition and then varying the fuel-air ratio from about 0.016 to 0.056. The higher value was at times limited by maximum liner temperature, facility exhaust-gas cooling capability, or an appreciable decrease in combustion efficiency.

The exhaust-gas thermocouples were used to obtain circumferential and radial temperature profiles and pattern factors. These thermocouples were removed for tests during which the exhaust-gas temperature would be greater than about 1750 K. In order to have consistent results, gas analysis data were used over the entire fuel-air ratio range for determining combustion efficiency. Arithmetical average liner metal temperatures were determined from 25 (maximum) thermocouples installed on the liners. There were 12 thermocouples on the inner and 13 thermocouples on the outer liner.

The fuel injection system consists of two circumferential rows of fuel modules. The fuel-flow rates of the two rows (inner and outer) could be varied individually by fuel-flow controls. Tests were conducted to determine if varying the flow split between the two rows, with other inlet conditions constant, would have an effect on combustor performance parameters.

## Results and Discussion

### Performance

*Combustion efficiency.*—Figure 8 shows combustion efficiency as a function of fuel-air ratio and inlet-air pressure and temperature. The inlet-air pressure variation was limited and had no apparent effect on the efficiency. However, the effects of varying inlet-air temperature and fuel-air ratio on combustion efficiency are much more pronounced. At the lower inlet-air temperature condition, combustion efficiency varied from 96 percent at a fuel-air ratio of 0.02 to 99 percent at 0.036 and then decreased to 94 percent as the fuel-air ratio was increased to 0.055. With the inlet-air temperature raised to 894 K, efficiency approached 100 percent from 0.016 to 0.045 fuel-air ratio and then decreased slightly as the fuel-air was increased to 0.053.

Results of tests where there was a variation in the fuel-flow splits between the inner row and outer row of fuel modules indicated that combustion efficiency was not affected.

*Pattern factor.*—The test data indicated that, although combustion efficiency was not affected by wide variations in the fuel-flow splits to the inner and outer rows, there was an appreciable change in pattern factor. As shown in figure 9, varying the flow split from about 27 to 52 (percentage of total fuel flow to the inner row) changed the pattern factor from 0.18 to 0.36. Pattern factor is defined as follows:

$$\frac{\text{Max. local exhaust temp.} - \text{Av. exhaust temp.}}{\text{Av. exhaust temp.} - \text{Av. inlet-air temp.}}$$

*Liner temperature.*—The effect of the fuel-flow splits on both the inner and outer combustor liners' differential temperatures is shown in figure 10. Liner differential temperatures were calculated by

subtracting the inlet-air temperature from the arithmetical average of the particular liner thermocouple readings. Increasing fuel-flow splits to the inner row decreased the difference between the inner and outer liner differential temperatures, and this is desirable from differential expansion considerations. However, as shown in figure 9, increasing the flow split to the inner row increased the pattern factor. From the data of figures 9 and 10 an operating fuel-flow split range of 39 to 43 was chosen for subsequent tests.

*Pressure loss.*—Inlet-air temperature and flow were varied without combustion to obtain the isothermal pressure loss of the combustion system and combustor liner. The system percent loss is calculated by

$$\frac{\text{Av. inlet total press.} - \text{Av. ex. total press.}}{\text{Av. inlet total press.}} \times 100$$

Liner percent loss is

$$\frac{\text{Av. liner annulus static press.} - \text{Av. ex. total press.}}{\text{Av. inlet total press.}} \times 100$$

Isothermal pressure loss data from both the combustion system and combustor liner are shown as a function of diffuser-inlet Mach number in figure 11. For example, at a diffuser-inlet Mach number of 0.35 the system isothermal loss was 5.4 percent, but the liner loss was only 0.9 percent. Most of the system loss was in the diffuser and in the turning of the flow path leading to the liner annuli.

*Unburned hydrocarbon emission index.*—Figure 12 shows the unburned hydrocarbon emission index, in grams of unburned hydrocarbon per kilogram of fuel, as a function of fuel-air ratio and inlet-air pressure and temperature. The narrow range of inlet-air pressures used for the tests had no appreciable effect on the index. For the 589 K inlet-air temperature condition (fig. 12(a)) an index value of 29 was attained at an 0.018 fuel-air ratio. The index value decreased to about 0.5 as the fuel-air ratio was increased to 0.045. Further increase in fuel-air ratio to 0.056 caused a rise in the index value to about 5.0.

Results of the tests at 894 K inlet-air condition (fig. 12 (b)) show the large effect that inlet-air temperature has on the unburned hydrocarbon emissions. An index value of 0.7 was obtained at a fuel-air ratio of 0.016; the index value decreased to less than 0.1 at a fuel-air ratio of 0.022 and remained less than 0.1 as the fuel-air value was further increased to 0.053.

*Carbon monoxide emission index.*—Figure 13 shows the carbon monoxide emission index, in grams of carbon monoxide per kilogram of fuel burned, as a function of fuel-air ratio and inlet-air pressure and temperature. As with the unburned hydrocarbon index, air pressure variation had a negligible effect on the carbon monoxide emissions. Figure 13(a) shows data taken at 589 K inlet-air temperature. The carbon monoxide index decreased from about 100 at 0.019 fuel-air ratio to a minimum of 30 at 0.034 fuel-air ratio. The index then increased to 300 as the fuel-air ratio was raised to 0.056. The dashed curve represents the equilibrium carbon monoxide concentration calculated from reference 7 for an inlet-air temperature of 589 K, ASTM Jet A fuel, and 0.505-MPa combustion pressure. This calculation predicts the minimum concentration of carbon monoxide possible for the given conditions. The difference between the data and the equilibrium curve is a measure of the inefficiency attributed to unburned carbon monoxide.

Tests at the 894 K inlet-air condition are shown in figure 13(b). The carbon monoxide emission index values are considerably lower at the higher inlet-air temperature over the fuel-air ratio range. An index value of 23 was attained at a 0.016 fuel-air ratio; as fuel-air ratio was increased to about 0.03, the index value dropped to 3. However, further increase in fuel-air ratio to 0.053 caused the index value to rise again to 150. The carbon monoxide equilibrium concentration for the 894 K inlet-air temperature is included in this figure. At the higher inlet-air temperature the difference between the measured carbon monoxide emission index and the equilibrium curve is much less, an indication that the inefficiency attributed to unburned carbon monoxide was small.

*Oxides of nitrogen emission index.*—Figure 14 shows the oxides of nitrogen emission index, in grams of nitrogen dioxide per kilogram of fuel burned, as a function of fuel-air ratio and inlet-air pressure and temperature. As with the unburned hydrocarbon index and the carbon monoxide index, the narrow range of inlet pressures used had no appreciable effect on the oxides of nitrogen index. Increasing the inlet-air temperature to 894 K from 589 K, at a 0.035 fuel-air ratio, resulted in a three- to fourfold increase in the oxides of nitrogen index values. As the fuel-air ratio was raised to 0.053, the emissions index values decreased to about 13.

*Smoke density.*—Only two measurements were made to determine the smoke density in the combustor exhaust gases. The tests were made at 0.505-MPa inlet-air pressure, 589 K inlet-air temperature, and fuel-air ratios of 0.026 and 0.042. A model 473A Engine Smoke Emission Sampler, manufactured by Robert Smith Electric Co., Inc., was used to obtain the stain on the Whatman number 4 filter paper. The

absolute reflectivity of the stained filter paper was measured with a Welsh Densichron using a gray background. The Densichron was calibrated with a Welsh Black Scale. The smoke index was determined from the following equation:

$$1 - \frac{\text{\% absolute reflectivity of sample}}{\text{\% absolute reflectivity of clean paper}} \times 100$$

Two smoke numbers were obtained: 3.6 at a 0.026 fuel-air ratio, and 10.3 at a 0.042 fuel-air ratio. Both values are considerably below the visible smoke region, the number for which is about 25.

Durability

*Liner temperatures.*—One parameter that can give some measure of the durability of the combustor liners is liner metal temperature. Figure 15 presents liner differential temperatures obtained with variations in inlet-air pressure and temperature and fuel-air ratio. The differential temperature is calculated by subtracting the average inlet-air temperature from the arithmetical average of all the liner thermocouple readings for any particular operating condition. Also included is the maximum local differential temperature taken from one of the thermocouples used for obtaining average temperatures. The data indicate a slight increase in liner differential temperature as combustor pressure was raised from 0.505 to 0.723 MPa. Liner average differential temperatures were as high as 320 K at 0.056 fuel-air ratio and 589 K inlet temperature. The maximum local temperature at this condition was 510 kelvins differential, or about 1100 K actual temperature. Comparison with the data taken at the higher inlet-air temperature, 894 K, shows similar values of liner differential temperature and maximum local differential temperature as a function of fuel-air ratio. At a fuel-air ratio of 0.053 an average differential temperature of 310 kelvins was obtained with a maximum differential temperature of 515 kelvins, or an actual temperature of 1409 K. This temperature is considerably greater than the maximum allowable of 1260 K used for thermal, stress, and film cooling analyses.

Since the tests were conducted at about 7 atmospheres pressure and 40 atomspheres pressure was used for design, no particular liner damage was caused by the higher-than-design metal temperature. Figure 5 shows the liner after 80 hours of operation. This included about 18 hours at exhaust-gas temperatures above 2150 K.

*Combustion gas temperatures.*—Figure 16 presents combustor exhaust-gas temperatures as a function of inlet-air pressure and temperature and fuel-air ratio. The temperatures were all calculated from gas analysis data. The narrow pressure range used in the tests showed no effect of pressure on the combustion gas temperatures. The data trend shown in figure 16 and the decrease in combustion efficiency shown in figure 8(a) indicate that a combustion gas temperature of 2250 to 2300 K is about the maximum attainable at the test pressures and 589 K inlet-air temperature. At 894 K inlet-air temperature a combustion gas temperature of 2400 to 2450 K could probably be attained.

## Summary of Results

The low-pressure performance was determined for a combustor designed to operate at high pressure and high exhaust-gas temperature. Data were taken at pressures to 7 atmospheres, inlet-air temperatures of 589 and 894 K, and exhaust-gas temperatures to 2365 K. The following results were obtained:

1. At an inlet-air temperature of 894 K the combustion efficiency was virtually 100 percent to fuel-air ratios of 0.045. At fuel-air ratios greater than 0.045 combustion efficiency decreased with increasing fuel-air ratio and was 98.5 percent at a fuel-air ratio of 0.053.

2. At an inlet-air temperature of 894 K and a fuel-air ratio of 0.053 the combustor exhaust-gas temperature was 2365 K.

3. At an inlet-air temperature of 894 K and an exit temperature of 2365 K the average of all combustor liner thermocouples indicated an average metal temperature of 1204 K (310 kelvins greater than the inlet-air temperature). The maximum metal temperature at this condition was 515 kelvins above the inlet-air temperature. No liner damage was observed after several hours of testing.

4. At a pressure of 5 atmospheres and a 589 K inlet-air temperature, smoke emissions increased with increasing fuel-air ratio. A smoke number of 10 was measured at a fuel-air ratio of 0.042.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, March 14, 1980,
505-04.

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TABLE I. - FILM-COOLING-AIR FLOW CALIBRATION OF COMBUSTOR LINERS

| Panel | Number of film holes per panel | Diameter of film holes, cm | Total film-hole area per panel, A, $cm^2$ | Calibration value of $AC_d$[a] | Calibration value of $C_d$ |
|---|---|---|---|---|---|
| | | | Inner liner | | |
| 1 (Upstream) | 142 | 0.282 | 8.865 | 0.0316 | 0.00357 |
| 2 | 132 | .295 | 9.000 | ------ | ------- |
| 3 | 132 | .318 | 10.451 | .0375 | .00359 |
| 4 | 132 | .295 | 9.000 | .0319 | .00354 |
| 5 | 142 | .239 | 6.358 | ------ | ------- |
| 6 | 162 | .226 | 6.502 | ------ | ------- |
| 7 (Downstream) | 178 | .170 | 4.049 | ------ | ------- |
| All panel holes unblocked | | | 54.225 $cm^2$/liner | 0.1866 | 0.00344 |
| | | | Outer liner | | |
| 1 (Upstream) | 220 | 0.249 | 10.706 | 0.0385 | 0.00360 |
| 2 | 252 | .239 | 11.283 | .0444 | .00394 |
| 3 | 258 | .253 | 12.943 | .0479 | .00370 |
| 4 | 228 | .254 | 11.553 | .0419 | .00363 |
| 5 | 220 | .218 | 8.245 | .0308 | .00374 |
| 6 | 208 | .226 | 8.348 | .0311 | .00373 |
| 7 (Downstream) | 196 | .170 | 4.458 | .0176 | .00395 |
| All panel holes unblocked | | | 67.536 $cm^2$/liner | 0.2488 | 0.00368 |

[a]Where $W = AC_d \sqrt{\rho \Delta P}$ and W is film-cooling-air flow in kg/sec, A is the total area of film-cooling-air holes unblocked during test in $cm^2$, $\rho$ is the density of air entering film-cooling-air holes in $kg/m^3$, and $\Delta P$ is pressure differential across liner in kPa.

TABLE II. - COMBUSTOR NOMINAL TEST CONDITIONS

| Combustor inlet | | | Mach number | | Film-cooling air | | | |
|---|---|---|---|---|---|---|---|---|
| Total pressure, MPa | Airflow rate, kg/sec | Total temperature, K | Diffuser inlet | Combustor reference | Flow rate,[a] kg/sec: Inner liner | Flow rate,[a] kg/sec: Outer liner | Flow, percent of combustor total: Inner liner | Flow, percent of combustor total: Outer liner |
| 0.505 | 10.70 | 589 | 0.319 | 0.030 | 0.65 | 0.89 | 6.1 | 8.1 |
| .578 | 12.23 | ↓ | ↓ | ↓ | .75 | .99 | ↓ | ↓ |
| .649 | 13.76 | ↓ | ↓ | ↓ | .84 | 1.11 | ↓ | ↓ |
| .723 | 15.29 | ↓ | ↓ | ↓ | .93 | 1.24 | ↓ | ↓ |
| .505 | 10.81 | 894 | .419 | .038 | .66 | .88 | ↓ | ↓ |
| .649 | 13.90 | 894 | .419 | .038 | .85 | 1.13 | ↓ | ↓ |

[a]Calculated from film-cooling-air flow equations previously noted in table I and isothermal $\Delta P$ values from fig. 11.

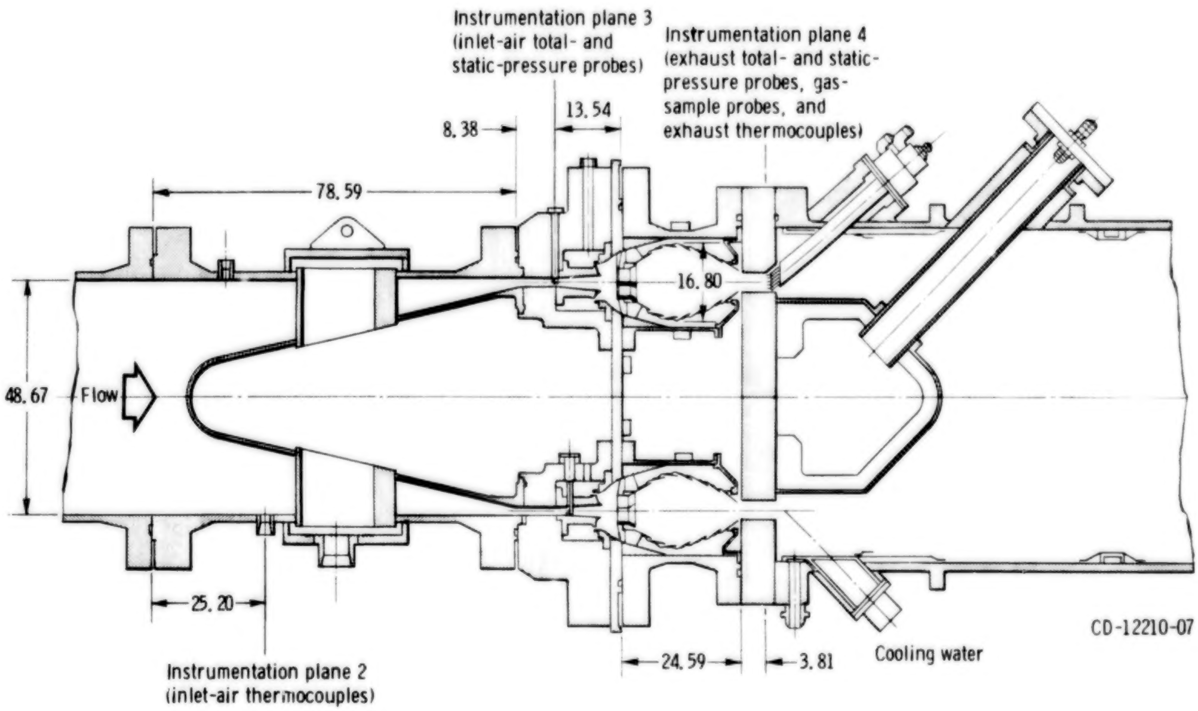



Figure 1. - Cross-sectional schematic of combustion test rig showing combustor liner test section. (Dimensions are in centimeters.)

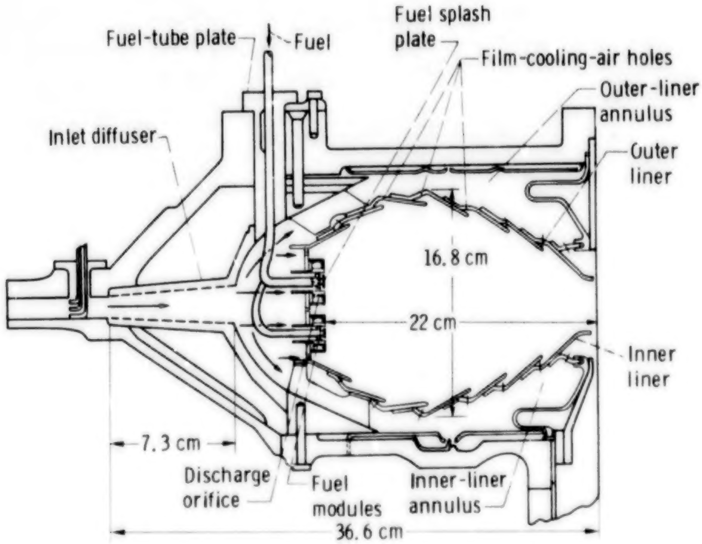
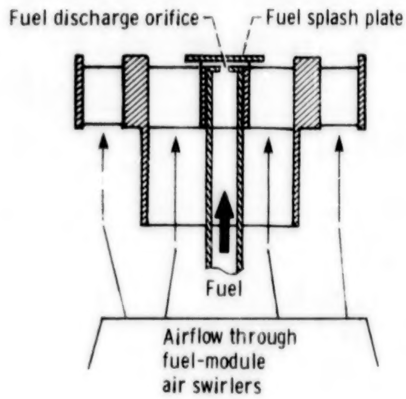
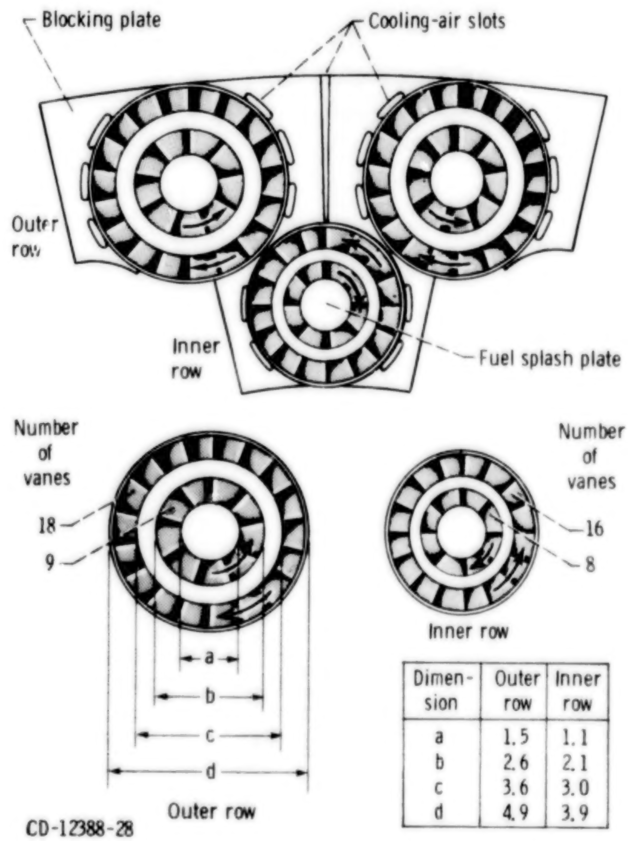



Figure 2. - Cross-sectional schematic of combustor showing inner and outer liners and fuel nozzles.




(a) View showing air and fuel flow passages.




| Dimen-sion | Outer row | Inner row |
|---|---|---|
| a | 1.5 | 1.1 |
| b | 2.6 | 2.1 |
| c | 3.6 | 3.0 |
| d | 4.9 | 3.9 |

(b) View looking upstream showing air swirlers and air discharge direction. (Dimensions are in centimeters.)

Figure 3. - Schematic of fuel modules.

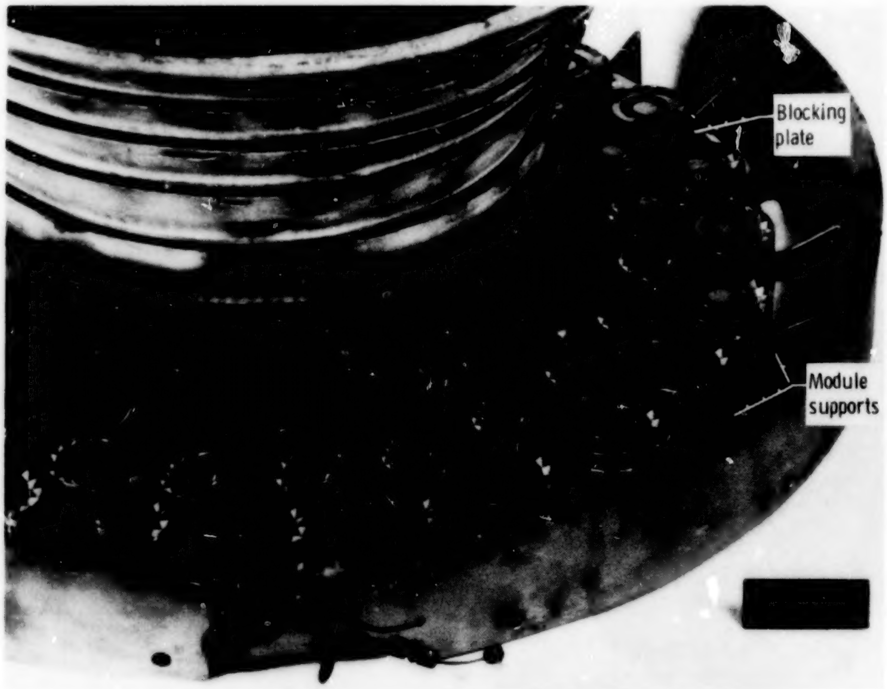



Figure 4. - Two rows of fuel modules, with inner combustor liner installed. (Also shown are fuel-module blocking plates and outer-row module supports.)

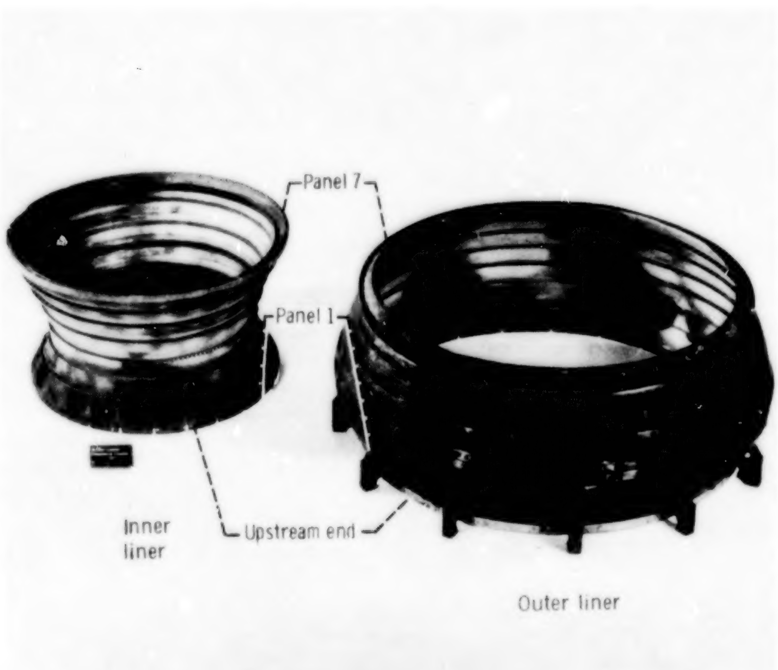



Figure 5. - Combustor liners after 80 hours of operation. (Thermocouple instrumentation has been removed.)

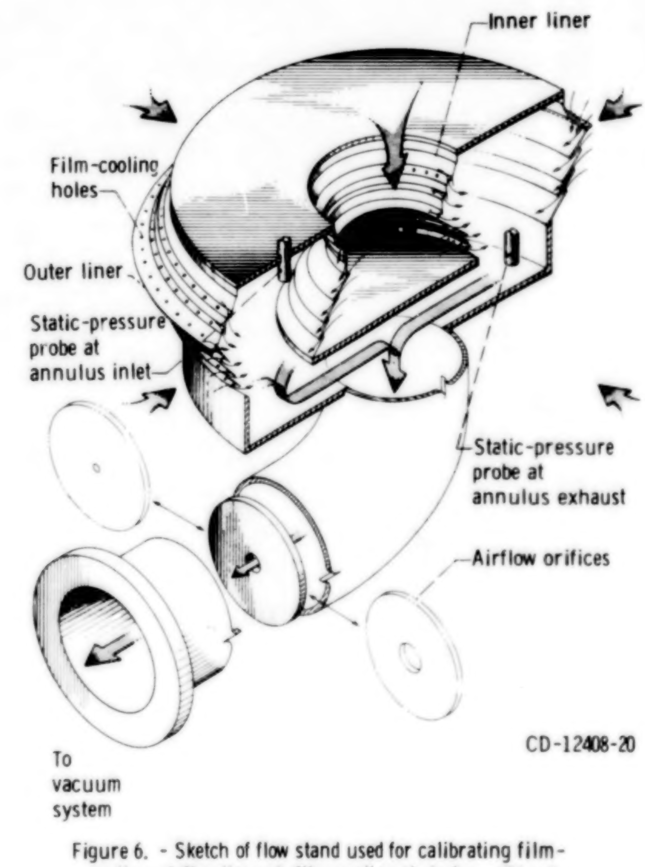
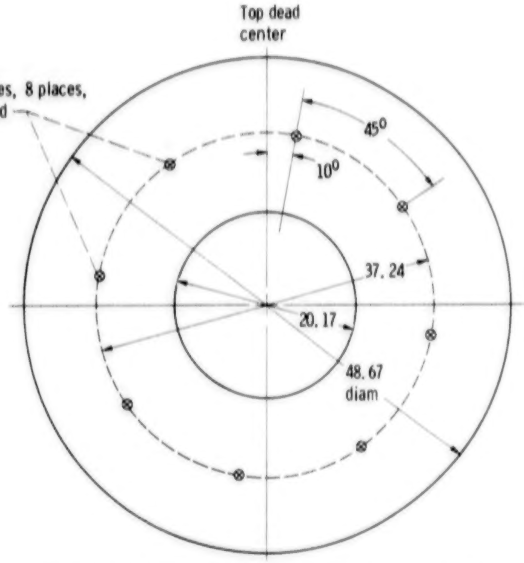
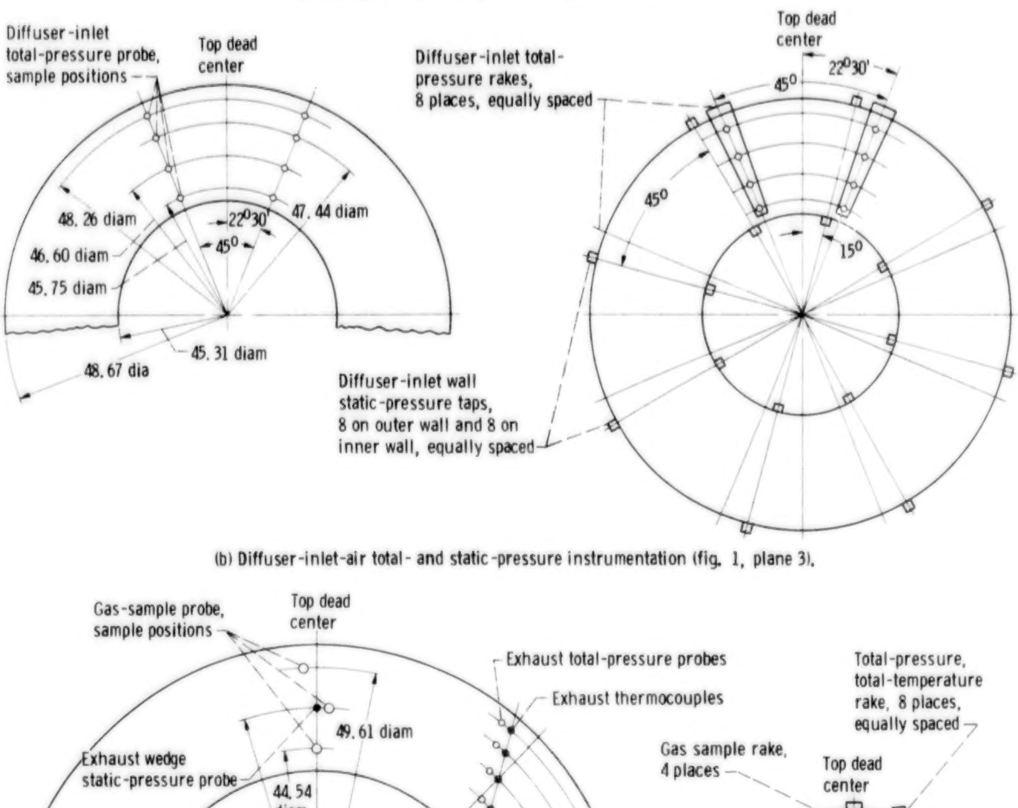


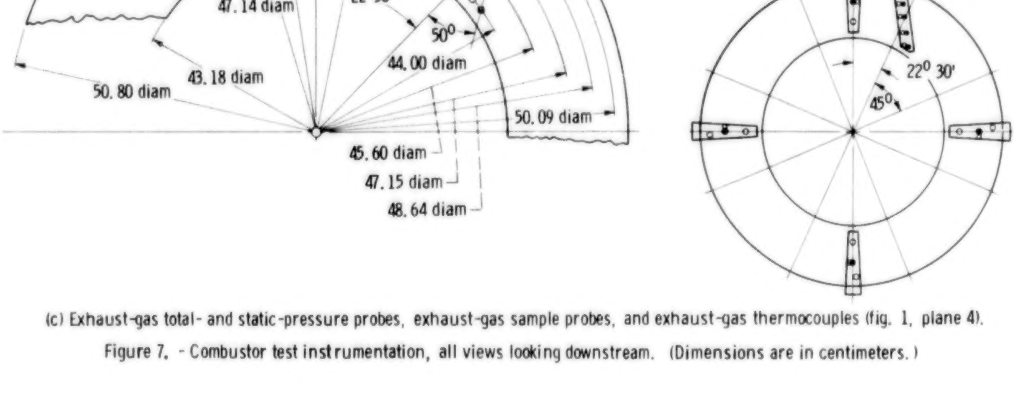

Figure 6. - Sketch of flow stand used for calibrating film-cooling-airflow through film-cooling air holes. (Single or multiple rows of holes can be calibrated.)




(a) Inlet-air Chromel-Alumel thermocouple (fig. 1, plane 2).




(b) Diffuser-inlet-air total- and static-pressure instrumentation (fig. 1, plane 3).




(c) Exhaust-gas total- and static-pressure probes, exhaust-gas sample probes, and exhaust-gas thermocouples (fig. 1, plane 4).

Figure 7. - Combustor test instrumentation, all views looking downstream. (Dimensions are in centimeters.)

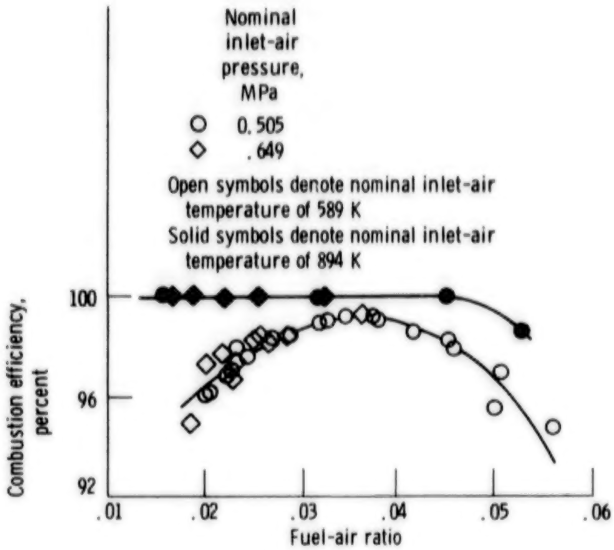



Figure 8. - Effect of fuel-air ratio on combustion efficiency at two combustor inlet-air pressures and two inlet-air temperatures.

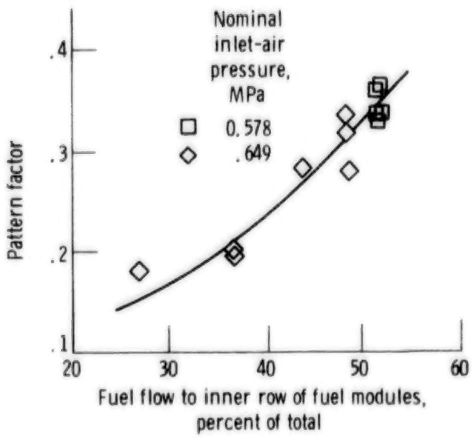



Figure 9. - Effect of fuel-flow split between inner row and outer row of fuel modules on combustion gas pattern factors at two combustor inlet-air pressures. Nominal inlet-air temperature, 589 K; fuel-air ratio, 0.0221.

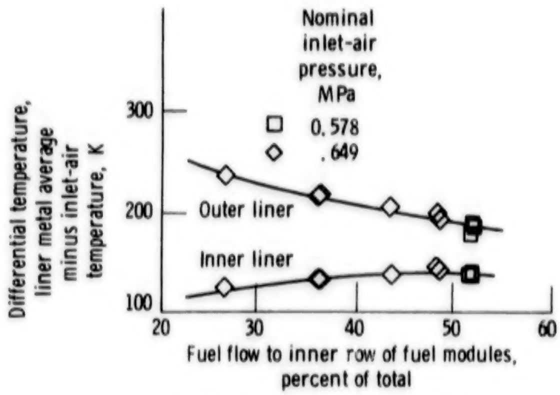



Figure 10. - Effect of fuel-flow split between inner row and outer row of fuel modules on combustor inner and outer liner average differential temperature at two combustor inlet-air pressures. Nominal inlet-air temperature, 589 K; fuel-air ratio, 0.0226.

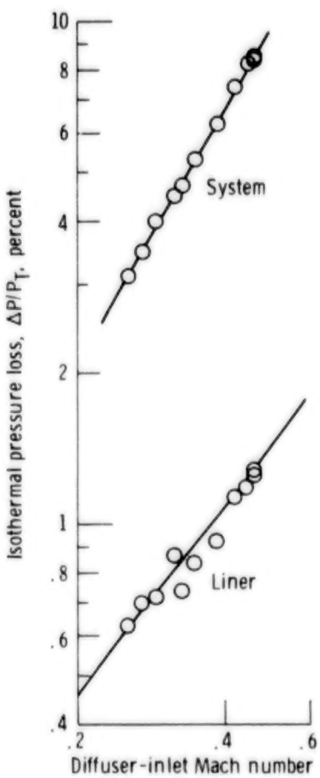
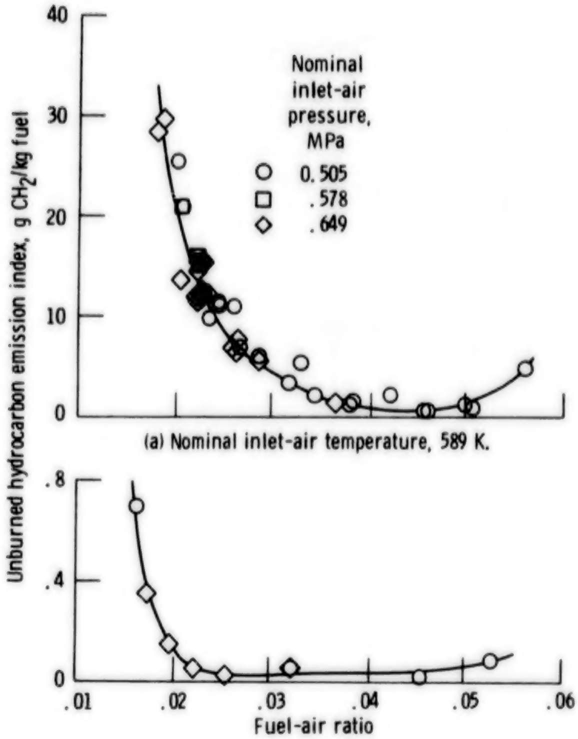



Figure 11. - Effect of diffuser-inlet Mach number on isothermal (nonburning) pressure loss across both entire combustion system and combustor liner only. Nominal inlet-air pressure, 0.505 MPa; variable airflow rate and temperature.




(b) Nominal inlet-air temperature, 894 K.

Figure 12. - Effect of fuel-air ratio on unburned hydrocarbon emission index at three combustor inlet-air pressures and two inlet-air temperatures.

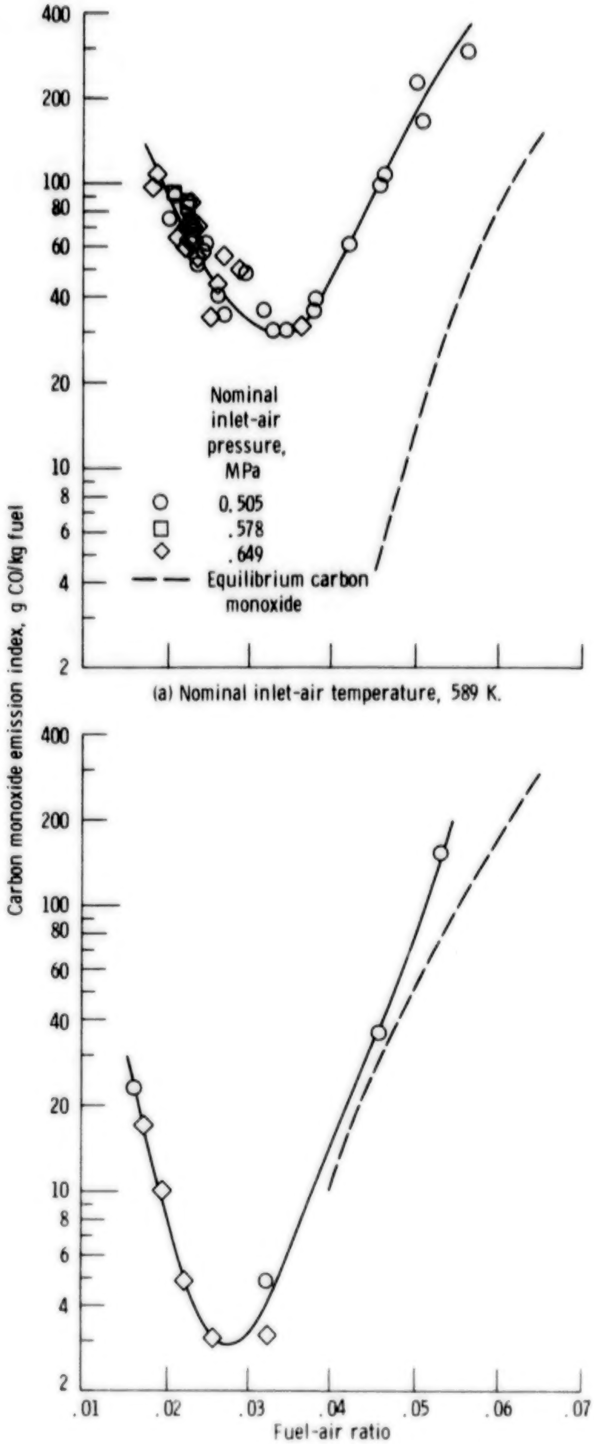



Figure 13. - Effect of fuel-air ratio on carbon monoxide emission index at three combustor inlet-air pressures and two inlet-air temperatures.

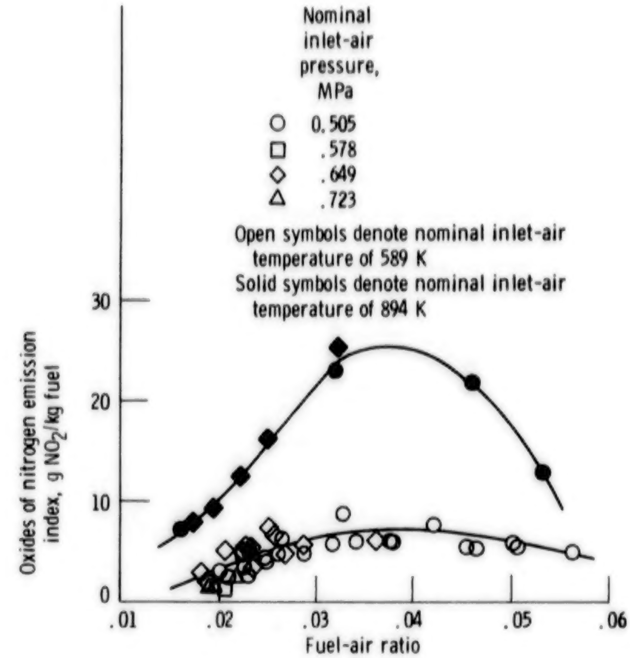



Figure 14. - Effect of fuel-air ratio on oxides of nitrogen emission index at four combustor inlet-air pressures and two inlet-air temperatures.

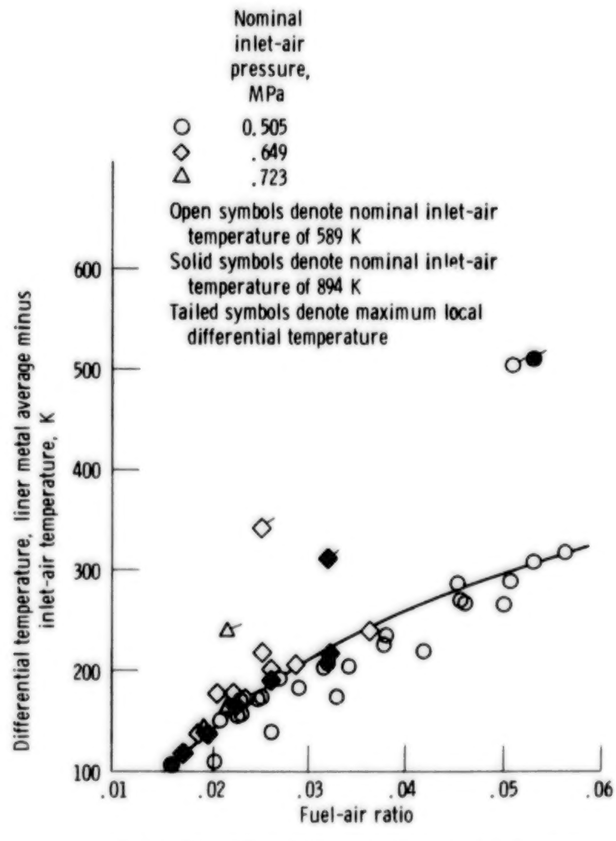



Figure 15. - Effect of fuel-air ratio on combustor-liner average differential temperature at three combustor inlet-air pressures and two inlet-air temperatures.

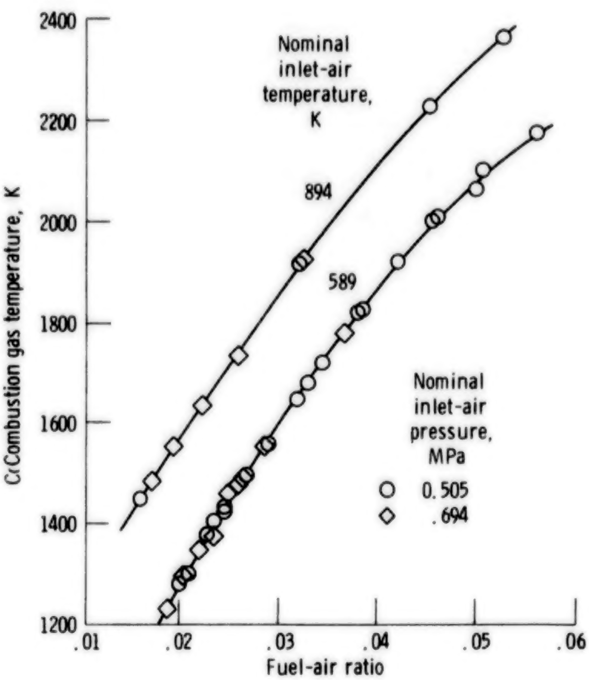



Figure 16. - Effect of fuel-air ratio on average combustion gas temperature at two combustor inlet-air pressures and two inlet-air temperatures. Gas temperatures were calculated from gas analysis data.

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16. Abstract

Experimental tests were conducted to develop a combustion system for a 40-atmosphere-pressure, 2480 K exhaust-gas-temperature turbine-cooling facility. The tests were conducted in an existing facility with a maximum pressure capability of 10 atmospheres and where inlet-air temperatures as high as 894 K could be attained. Exhaust-gas temperatures were as high as 2365 K. Combustion efficiencies were about 100 percent over a fuel-air ratio range of 0.016 to 0.056. Combustion efficiency decreased at leaner and richer ratios when the inlet-air temperature was 589 K. Data are presented that show the effect of fuel-air ratio and inlet-air temperature on liner metal temperature. Isothermal system pressure loss as a function of diffuser-inlet Mach number is also presented. Data included exhaust-gas pattern factors; unburned hydrocarbon, carbon monoxide, and oxides of nitrogen emission index values; and smoke numbers.

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